GLEX-2102.D1.2.8x12585

TIME AND ENERGY, EXPLORING TRAJECTORY OPTIONS BETWEEN NODES IN EARTH-MOON SPACE

Author: Mr. Roland Martinez
NASA, United States, roland.m.martinez@nasa.gov

Mr. Gerald Condon
NASA, United States, gerald.l.condon@nasa.gov

Mr. Jacob Williams
ERC, Inc., Engineering and Science Contract Group, United States, jacob.williams@escg.jacobs.com

Abstract

The Global Exploration Roadmap (GER) was released by the International Space Exploration Co-ordination Group (ISECG) in September of 2011. It describes mission scenarios that begin with the International Space Station and utilize it to demonstrate necessary technologies and capabilities prior to deployment of systems into Earth-Moon space. Deployment of these systems is an intermediate step in preparation for more complex deep space missions to near-Earth asteroids and eventually Mars. In one of the scenarios described in the GER, "Asteroid Next", there are activities that occur in Earth-Moon space at one of the Earth-Moon Lagrange (libration) points. In this regard, the authors examine the possible role of an intermediate staging point in an effort to illuminate potential trajectory options for conducting missions in Earth-Moon space of increasing duration, ultimately leading to deep space missions.

This paper will describe several options for transits between Low Earth Orbit (LEO) and the libration points, transits between libration points, and transits between the libration points and interplanetary trajectories. The solution space provided will be constrained by selected orbital mechanics design techniques and physical characteristics of hardware to be used in both crewed missions and uncrewed missions. The relationships between time and energy required to transfer hardware between these locations will provide a better understanding of the potential trade-offs mission planners could consider in the development of capabilities, individual missions, and mission series in the context of the ISECG GER.

I. INTRODUCTION

The early phase of the GER[1] Asteroid Next mission scenario begins in the 2022-2023 timeframe with the deployment of a core Deep Space Habitat (DSH) in cis-Lunar space. The first version of the GER suggested this module be delivered to Earth-Moon Lagrange point 1 (EM-L1). This would allow for demonstration of long duration habitation and other critical systems and enable a series of progressively longer duration human missions in a deep space environment as reduction of dependence on a regular supply chain from Earth is realized. These early missions are conducted until the additional capabilities required to conduct the Near Earth Asteroid (NEA) exploration missions are fully developed and available. This represents a risk reduction approach in cis-Lunar space before committing humans to long duration missions with limited mission abort capabilities. In this scenario, abort times to return to Earth are measured in only a few days. Missions of increasing duration are described that could be conducted during this timeframe,

which are not restricted to one of the Lagrange points. These missions may use a Lagrange point as an anchor (specifically Earth-Moon L2) for a proof of concept and for assessment of longer duration transits and crew operations within a relatively safe distance from Earth.

II. HALO ORBIT MISSION DESIGN CONSIDERATIONS

Halo Orbits

A halo orbit* about the Earth-Moon L2 Lagrange point (EM-L2) can serve as an intermediate location for spacecraft to execute missions to other destinations[2]. There are two halo families, north and south, which reflect whether the maximum z-amplitude magnitude on the north or south side of the Earth-Moon plane. The halo amplitude is a measure of the excursion of the halo from the Earth-Moon plane (the y-axis in Figure I).

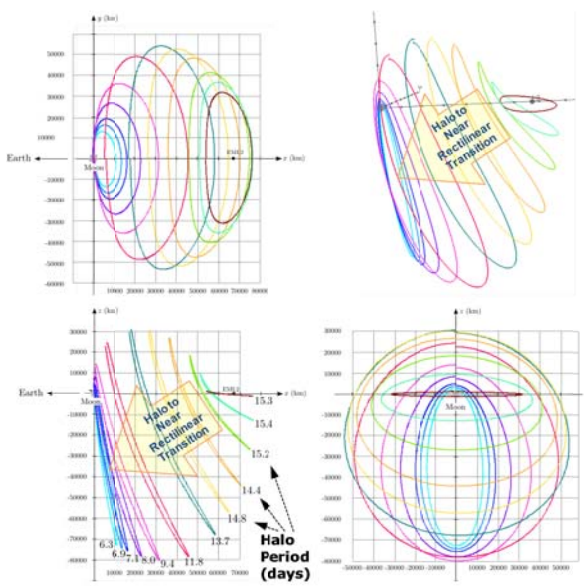


Fig. I Earth-Moon L2 Lagrange point south family. The is at the origin of the coordinate frame.

* The term "orbit" describes the apparent motion of a particle in a selected halo in a rotating frame of reference.

Lower z-amplitude halos tend to be located around the L2 point. As the magnitude of the amplitude is increased, the halo rotates out of the Earth-Moon plane while, at the same time, migrating from the L2 location toward the moon. Halo amplitudes range from approximately 0 to 30,000 km for the north family and up to 80,000 km for the south family. The period of these orbits are about 14-15 days, or approximately half the period of the lunar orbit about the Earth. "Near rectilinear halos" reside near the moon and have a shorter period of about 6-7 days. The lunar closest approach of a near rectilinear halo, for example, can be selected, but then the z-amplitude of the farthest point of the orbit is then determined. While all of the halo orbits are unstable, the near rectilinear halo is less unstable than the halos closer to the L2 point. The near rectilinear halos have been suggested as a candidate orbit for a mission to monitor a particular region of the moon (e.g., the south pole) due to the bulk of the orbit spending time at some elevation above the south pole. The lower z-amplitude halos, while more unstable, are more amenable to transition to and from weak stability boundary (manifold) trajectories.

L2 or L2 halo orbit selection will be influenced by transportation related considerations such as flight times and delta-V (ΔV) requirements. Additionally, halo orbit selection may depend on non-transportation related parameters:

- Environmental considerations in L2 halo orbits – thermal, radiation, micro-meteoroid and orbital debris
- Orbit maintenance and rendezvous costs
- Earth communication and line of sight visibility
- Lunar site visibility (e.g., south pole)
- Science objectives

Direct, Flyby, and Low Energy (Manifold) Trajectory)

There are a number of possible outbound (i.e., LEO to EM-L2) trajectories that could carry a crew or cargo from LEO to a Lagrange point halo. Three shown in Figure II are Direct, lunar

Flyby, and a low energy, manifold trajectory transfer.

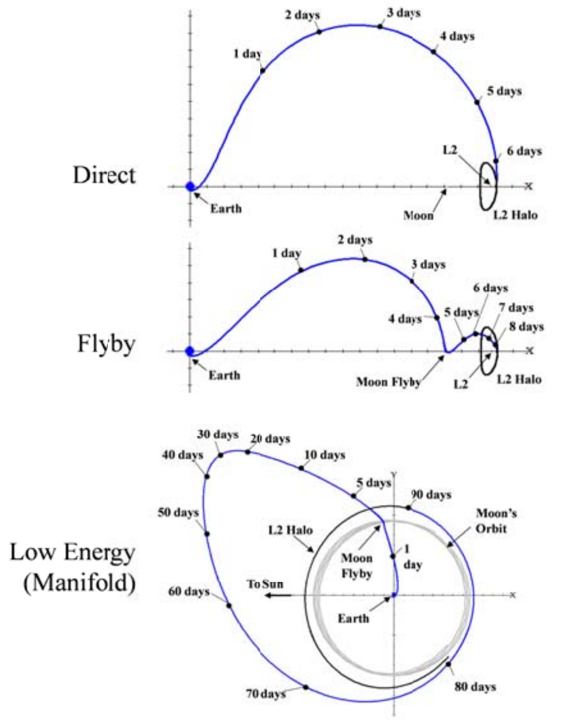


Fig. II LEO to Earth-Moon L2 halo trajectories for manned and cargo flights.

A Direct transfer is the fastest of the three flight techniques shown here with a LEO to EM-L2 flight time of approximately 6-7 days. For this case, the trajectory, which supports a manned mission, would carry the crew from LEO to a transfer trajectory that arrives at the EM-L2 halo insertion location. This fast trajectory comes at the cost of the highest total ΔV cost and would be considered for a manned mission (see Table I for a comparative list of example cases).

The introduction of a lunar flyby causes a significant reduction in the post Earth-departure (i.e., Lunar Flyby plus L2 arrival) ΔV at the expense of a modest increase in flight time to 8-10 days. This trajectory begins with a LEO departure to a powered lunar flyby and ending with an insertion maneuver into an EM-L2 halo. In this example case, the Lunar Flyby technique reduces the lunar flyby and L2 halo arrival ΔV to about 30% that required for the Direct trajectory. This technique also supports a manned LEO to EM-L2 trajectory with still reasonable flight times.

The low energy (Manifold) trajectory essentially eliminates the lunar flyby and EM-L2 halo arrival ΔV requirement. This flight technique applies to a cargo mission, but not to a manned mission due to the long flight time. The trajectory departs LEO and, near the Earth's sphere of influence, enters without any powered maneuver onto a manifold trajectory that continues on to coast to an unpowered EM-L2 halo insertion.

Initial 185x185 km LEO Altitude					
Mission Type	Flight Time (days)	Earth Departure C3 (km^2/s^2)	LEO Departure ΔV (m/s)	L2 Halo Arrival + Flyby ΔV (m/s)	Total ΔV (m/s)
Direct	6.3	-1.685	3151	967	4118
Lunary Flyby	8.4	-2.083	3133	284	3417
Manifold	89.6	-1.991	3195	0	3195

Table I: LEO to EM-L2 halo ΔV example requirements for Direct, Lunar Flyby, and Manifold insertion trajectories.

III. CANDIDATE MISSIONS

This assessment examines nominal ΔV requirements for selected candidate exploration missions that travel in the vicinity of the Earth-Moon and Sun-Earth systems. For some cases, associated nominal ΔV and mission time of flight (TOF) sensitivities are examined.

I. Overall Mission

The overall mission architecture begins with the launch of a free-flying module (hereafter referred to as the Core module) onto a low energy (e.g., manifold) trajectory to an EM-L2 halo. The Core module would serve as a habitable volume for the crew. It would also possess a propulsion system. In this case, the post-Earth departure portion of the ascent trajectory would carry the Core module through an unpowered lunar flyby and to a transition to

an Earth-Moon manifold that would ultimately deliver the spacecraft to an EM-L2 halo.

Subsequent to the launch of the Core module, a manned Multi-Purpose Crew Vehicle (MPCV; aka Orion) would be launched on a lunar flyby trajectory to rendezvous and dock with the Core module, already in the EM-L2 halo. The Orion vehicle currently possesses a maximum active lifetime of 21 days with a crew of 4[*]. Once docked with the Core module, the Orion vehicle becomes inactive in the sense that the crew can access resources from the Core module and so docked time does not count against Orion active lifetime.

Orion Capability

The time spent during the rendezvous and docking flight phase is about 10 days (approximately 8.5 days for launch to EM-L2 halo arrival and 1.5 days for rendezvous and docking). This allows Orion, with its 21 days of active crewed lifetime, up to 11 days for nominal or abort Earth returns (see Table II). Given an Orion spacecraft with a total ΔV capability of 1370 m/s, a combined outbound and rendezvous (post Earth departure) ΔV requirement of 360 m/s leaves 1010 m/s for nominal and abort Earth returns for the Orion.

MPCV	TOF (days)	ΔV (m/s)
Capability	21	1370
Outbound to EM-L2	-8.5	-350
Rendezvous	-1.5	-10
Abort Capability	11	1010

Table II: Orion time of flight and ΔV capability.

After docking and checkout, the Orion/Core stack is poised to perform an expeditionary mission or excursion to a selected destination or region. It can perform different types of missions: 1) a roundtrip mission where the Orion/Core stack travels to its destination target and then back to EM-L2, after which time the crew returns to Earth in the Orion spacecraft and 2) a one-way trip where the Orion/Core stack travels to its destination and the Orion returns the crew directly back to Earth from this destination, while the Core module independently returns to an EM-L2 halo.

II. 30 Day Class Mission: One Way Excursion to EM-L2 South Family

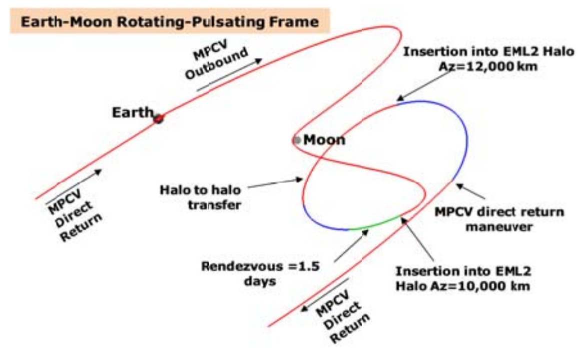


Fig. III 30-Day class mission. Earth-Moon L2 amplitude change for south family. Earth-Moon rotating pulsating frame.

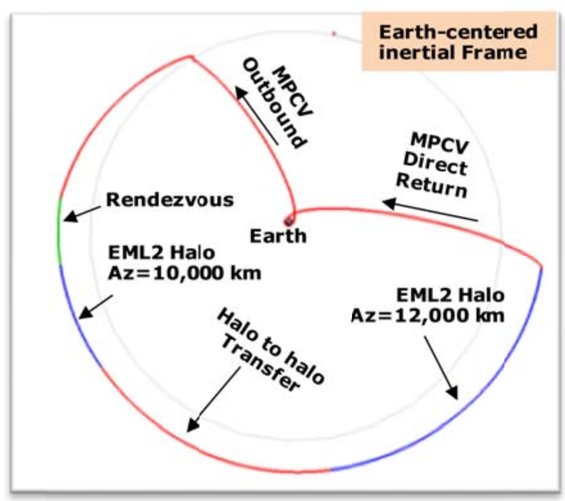


Fig. IV 30-Day class mission. Earth-Moon L2 amplitude change for south family. Earth-centered inertial frame.

Example mission 1 (shown in Figures III and IV) is an EM-L2 to EM-L2 halo transfer with a 30-day class duration (launch to landing). It represents an early low ΔV cost mission

[*] The maximum lifetime is driven by food stuffs that can be stored in the command module as opposed to consumables such as oxygen for breathing.

remaining in the general EM-L2 vicinity. This "shakedown" mission, which involves a transfer from one EM-L2 halo z-amplitude to another one, can be used to validate flight techniques, guidance, navigation, and control (GN&C) as well as other flight and environmental systems for subsequent and longer manned missions.

This mission begins with an Orion launch to a 28.5°inclination, 185x185 km altitude LEO phasing orbit, which provides an optimal location for an upper stage to place Orion on an Earth departure trajectory. The 185 km circular orbit altitude is a common reference for all missions. The 3158 m/s ΔV Earth departure* targets an EM-L2 halo via a 228 m/s powered lunar flyby, which achieves a 100 km closest approach to the Moon. About 8.6 days after Earth departure, Orion performs a 112 m/s ΔV maneuver, which inserts it onto an EM-L2 halo with a z-amplitude of 10,000 km. The Orion then takes 1.5 days to complete its active rendezvous, proximity operations, and docking with an awaiting Core module.

The docked Orion/Core spacecraft stack remains in the L2 halo for about 2.3 days during which time the crew performs transition of primary operations from Orion to the Core module spacecraft, followed by a checkout period prior to the halo to halo transfer mission. The Core module then performs a series of ΔV maneuvers totaling 16 m/s that take the stack from an EM-L2 halo z-amplitude of 10,000 km to 12,000 km. The entire transfer takes about 4.7 days.

After about 5.7 days in the new 12,000 km z-amplitude EM-L2 halo, Orion separates from the Core module and performs a 944 m/s ΔV maneuver, returning it to Earth entry interface (EI) altitude in about 5.9 days.

* Note that a capable launch vehicle that can lift the LEO payload to an elliptical departure parking orbit can reduce the LEO departure ΔV requirement. For example, a properly aligned 1806x185 km altitude LEO can reduce the LEO departure to 2,741 m/s (a savings of 417 m/s).

Subsequent to the Orion return to Earth, the Core module returns to EM-L2 on a slower, low energy trajectory (not shown). It remains there awaiting resupply and a new crew for future missions. The ΔV and flight time summary for example mission 1 is shown in Table III. The total mission duration is 28.7 days (16 days in the Orion, and 12.7 days on the Core/Orion stack).

Maneuver	**ΔV (m/s)**
Earth Departure	3158
Lunar Flyby	228
L2 Halo Insertion	112
L2 to L2	16
L2 Departure	944

Mission Phase	**Flight Time (days)**
LEO to L2 Halo	8.6
L2 Halo Rendezvous	1.5
Stay in L2 Halo	2.3
L2 to L2 Transfer	4.7
Stay in L2 Halo	5.7
L2 to Earth Return	5.9
Total	28.7

Table III: ΔV and Flight Time Summary for Mission Example 1.

III. Mission 2 - 60 day class – EM-L2 Halo to EM-L1 Halo

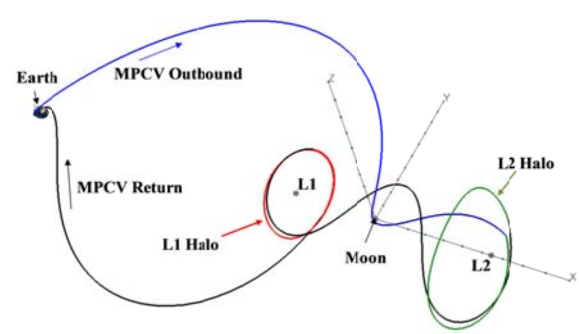


Fig. V Round trip MPCV Trajectory for Mission Example 2 in an Earth-Moon Rotating-Pulsating Frame.

Example mission 2 is a transfer from an L2 halo orbit to an L1 halo orbit. Figure V shows the round trip trajectory for this case. The mission for Orion starts at LEO where the 3138 m/s Earth departure ΔV maneuver is performed, placing the spacecraft on a 7.9-day transfer trajectory which includes a 169 m/s ΔV powered flyby of the Moon. Orion then inserts into a 21,096 km z-amplitude EM-L2 halo orbit with a 159 m/s ΔV maneuver and then spends 1.5 days to rendezvous and dock with the awaiting Core module. Note that the Earth departure, lunar flyby, and L2 halo insertion ΔVs are different than for mission example 1. This is primarily due to the differences in mission epoch and the associated differences in Earth-Moon geometry. The arrival L2 z-amplitudes are also different, due to the fact that they are allowed to vary in order to minimize the overall outbound ΔVs.

After coasting in L2 halo orbit for close to 1 revolution (about 13.5 days), the Core/Orion stack leaves the L2 halo orbit on an outbound manifold which takes it to the L1 halo inbound manifold (via an 18 m/s ΔV maneuver performed near the Moon). After staying in the L1 halo orbit for about 15.5 days, the Orion undocks from the Core module, and a final Orion maneuver takes place to return the crew to Earth via a direct transfer. The Core module returns to the L2 halo on a slower low-energy transfer (not shown). The ΔV and flight time summary for mission 2 is shown in Table IV. The total mission duration is 60 days (12 days in the MPCV, and 48 days in the Core/Orion stack).

Aborts from this trajectory were also examined. At various points along the mission, starting at arrival onto the L2 halo, both direct (single maneuver) and flyby (two maneuvers) return trajectories were computed. These results are shown in Figure VI, where the abort transfer time has been limited to no more than 11 days. In general, the flyby return will be cheaper, although there are areas in the vicinity of EM-L1 where a direct return is cheaper. It is noted that, at all points in the mission, there is a return option that is within the Orion vehicle capability.

Maneuver	ΔV (m/s)
Earth Departure	3138
Lunar Flyby	169
L2 Halo Insertion	159
L2 to L1 Midcourse	18
L1 Departure	546

Mission Phase	Flight Time (days)
LEO to L2 Halo	7.9
L2 Halo Rendezvous	1.5
Stay in L2 Halo	13.5
L2 to L1 Transfer	19.5
Stay in L1 Halo	15.5
L1 to Earth Return	4.6
Total	62.4

Table IV: ΔV and Flight Time Summary for Mission Example 2.

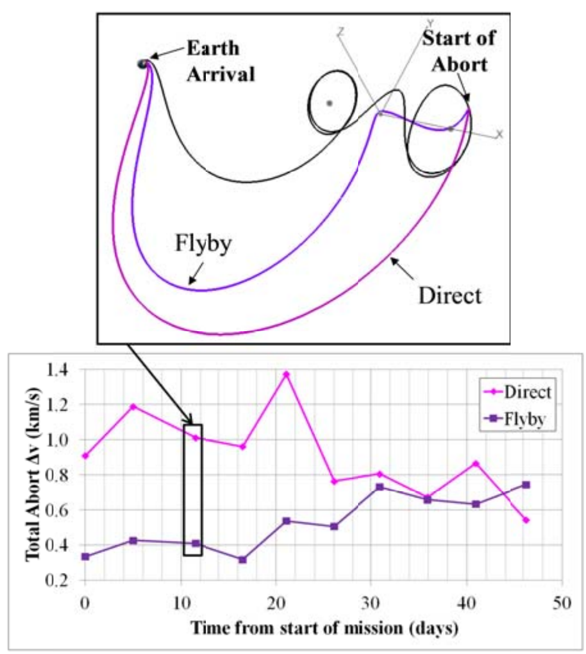


Fig. VI Direct and Flyby Aborts for Mission Example 2.

IV. Mission 3 - 90 day class – EM-L2 Halo to Trans-Earth to EM-L2 Halo

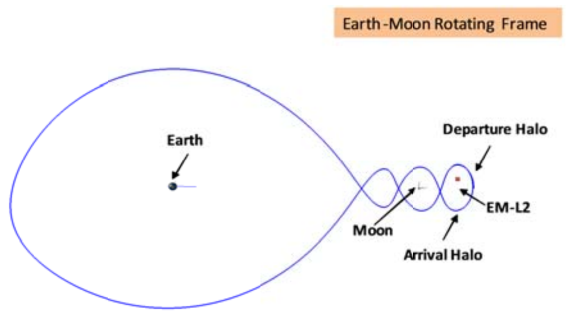


Fig. VII Round trip trans-Earth excursion for Mission Example 3 in an Earth-Moon Rotating Frame.

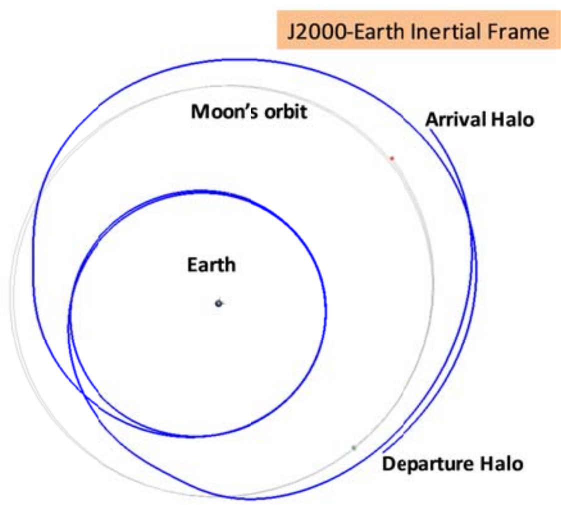


Fig. VIII Round trip trans-Earth excursion for Mission Example 3 in a J2000-Earth Inertial Frame.

Example mission 3 is a transfer from an EM-L2 halo orbit through a circuitous route around EM-L1, the moon, and trans-Earth; then back around the moon and L1 to completion at L2. Figures VII and VIII show the round trip trajectory for this case in an Earth-Moon Rotating Frame and a J2000-Earth Inertial Frame, respectively.

The Orion LEO to L2 mission phase to docking with the Core spacecraft at L2 is similar to that in mission 1, except that the L2 halo target z-amplitude for this mission is 10,400 km. After a brief coast in the L2 halo orbit, the Core/Orion stack performs a 1 m/s ΔV maneuver to depart L2 onto an outbound manifold which takes it on a tour of the Earth-Moon system. The stack departs L2 and circles about the moon, then about L1 followed by an extended passage around Earth including passing through the region of the trans-Earth L3 Lagrange point. As the tour continues, the trajectory takes the spacecraft back around L1, then around the moon and back to a 10,600 km z-amplitude L2 halo. After a short coast in the L2 halo, the Orion undocks from the Core module and performs a direct return to Earth at a ΔV cost of 944 m/s and a flight time of 5.9 days. The Core module remains in the L2 halo waiting to be resupplied for a future mission. The ΔV and flight time summary for mission 3 is shown in Table V. The total mission duration is approximately 91 days (16 days in the Orion, and 75 days in the Core/Orion stack).

Maneuver	ΔV (m/s)
Earth Departure	3158
Lunar Flyby	228
L2 Halo Insertion	112
L2 to Trans-Earth to L2	1
L2 Departure	944

Mission Phase	Flight Time (days)
LEO to L2 Halo	8.6
L2 Halo Rendezvous	1.5
Stay in L2 Halo	5.0
L2 to Trans-Earth to L2	65.0
Stay in L2 Halo	5.0
L2 to Earth Return	5.9
Total	91.0

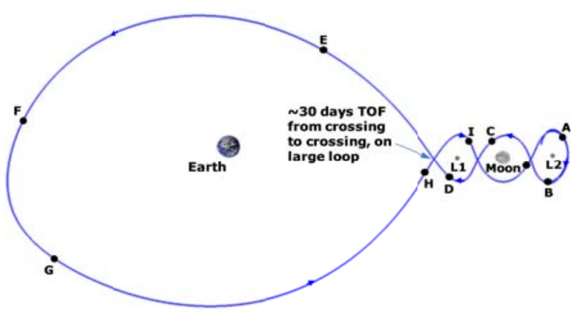
Table V: ΔV and Flight Time Summary for Mission Example 3.

The trajectory ΔV cost for the L2 excursion about the moon, L1, Earth, and ultimately back to L2 is sensitive to time of flight. Table VI shows that the overall mission duration could be reduced 20 days by increasing the L2 to Trans-Earth to L2 excursion ΔV from the current 1m/s to 82 m/s.

Time of Flight (days)	ΔV (m/s)
65	**1**
60	12
55	18
50	32
45	82

Table VI: ΔV and Flight Time Sensitivities for Mission Example 3, L2 to Trans-Earth to L2 flight phase.

Aborts from this trajectory were also examined. At various points along the mission, abort ΔV costs are noted starting at L2 halo departure. These results are shown in Figure IX, where the abort transfer time has been limited to no more than 11 days.



	No Moon Flyby		Moon Flyby	
Point	**TOF (days)**	**ΔV (m/s)**	**TOF (days)**	**ΔV (m/s)**
A			**10.2**	**299**
B			**5.7**	**354**
C	**4.8**	**570**	**4.8**	**1151**
D	**3.2**	**799**	**6.1**	**683**
E	**1.9**	**1020**		
F	**2.9**	**1005**		
G	**2.0**	**992**		
H	**3.4**	**927**		
I	**4.6**	**592**	**9.2**	**875**
J	**7.6**	**1236**	**7.3**	**489**

Fig. IX Abort locations and ΔV cost for selected points along the trajectory of mission 3.

It is noted that, at all points in the mission, there is a return option that is within the Orion vehicle capability. The time of flight for abort trajectories range from 1.9 to 10.2 days and the abort ΔVs range from 300 to 1226 m/s. Assuming that only 1010 m/s can be applied to either nominal or abort maneuvers (per Table II), then there are some abort maneuvers that will not be achievable, leaving ranges of time during this mission when an abort trajectory is not possible (within the 11 day return time limit).

Finally, for missions employing solar power, there exists particular interest in the solar eclipsing that occurs during the mission. For the mission 3 trajectory, there is one instance of a 3-hour eclipsing period. During this time, solar panels would experience either a full or partial eclipse. For such occurrences, backup power systems (e.g., batteries) would be required.

V. Mission 4 - 120 day class – EM-L2 Halo to (Near Rectilinear) EM-L2 Halo to EM-L2 Halo

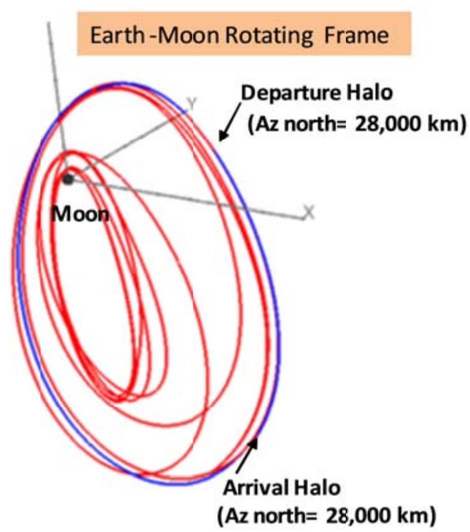


Fig. X Round trip mission to near rectilinear halo from L2 for Mission Example 4 in an Earth-Moon Rotating Frame.

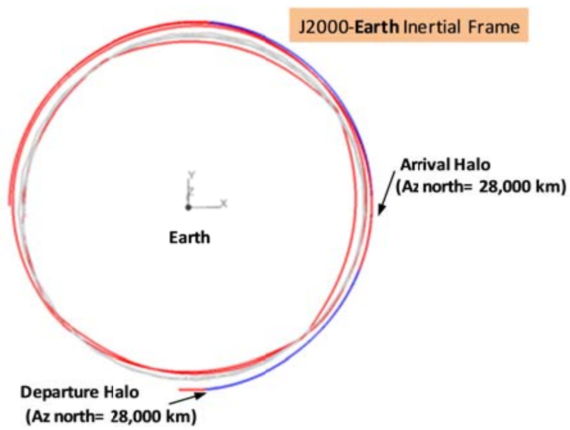


Fig. XI Round trip mission to near rectilinear halo from L2 for Mission Example 4 in a J2000-Earth Inertial Frame.

Example mission 4 is a transfer from an EM-L2 halo orbit to a near rectilinear (NR) halo and then back to EM-L2. Figures X and XI show the round trip trajectory for this case in an Earth-Moon Rotating Frame and a J2000-Earth Inertial Frame, respectively.

The Orion LEO to L2 mission phase to docking with the Core spacecraft at L2 is similar to that in mission 1, except that the L2 halo target z-amplitude for this mission is 28,000 km. After a coast of about 7 days, the Core/Orion stack departs L2 to a NR halo target with an 866 km closest approach to the moon. After a 31-day loiter in the NR halo, the stack performs a NR halo departure returning the crew to the 28,000 km z-amplitude halo in 25 days. The 79 m/s total ΔV cost for this sequence consists of 6 individual maneuvers: L2 halo departure, mid-course correction, L2 NR halo arrival, L2 NR departure, mid-course correction, and L2 halo arrival. After about 7 days back in the L2 halo, the Orion spacecraft separates from the Core module and performs a 944 m/s Earth return ΔV maneuver taking the crew to Earth entry interface in about 5.9 days. The Core module remains in the L2 halo. The ΔV and flight time summary for mission 4 is shown in Table VII. The total mission duration is approximately 106 days (16 days in the Orion, and 90 days in the Core/Orion stack).

Maneuver	ΔV (m/s)
Earth Departure	3158
Lunar Flyby	228
L2 Halo Insertion	112
L2 to NR to L2	79
L2 Departure	944

Mission Phase	Flight Time (days)
LEO to L2 Halo	8.6
L2 Halo Rendezvous	1.5
Stay in L2 Halo	7.0
L2 to NR Halo	31.0
NR Stay	20.0
NR to L2	25.0
Stay in L2 Halo	7.0
L2 to Earth Return	5.9
Total	106.0

Table VII: ΔV and Flight Time Summary for Mission Example 4.

VI. Mission 5 - 180 day class – EM-L2 Halo to Sun-Earth L2 Region to EM-L2 Halo

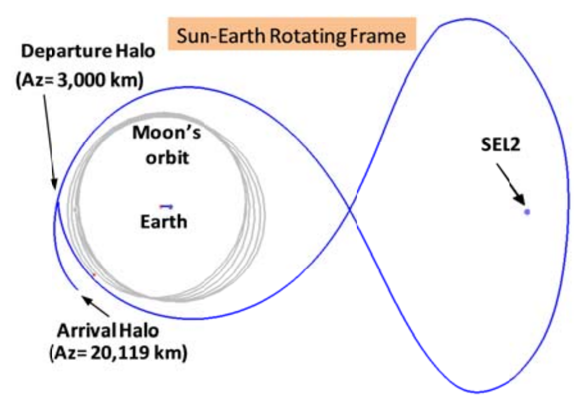


Fig. XII Round trip mission to Sun-Earth L2 region from EM-L2 for Mission Example 5 in a Sun-Earth Rotating Frame.

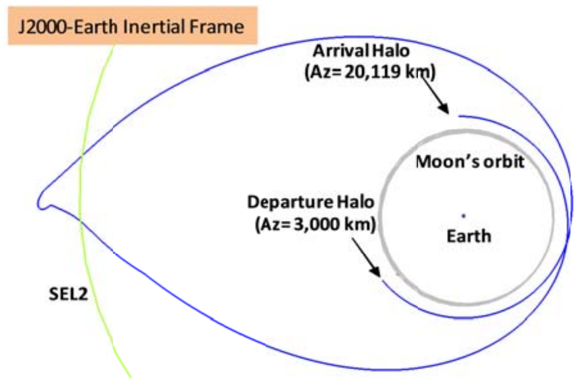


Fig. XIII Round trip mission to Sun-Earth L2 region from EM-L2 for Mission Example 5 in a J2000-Earth Inertial Frame.

Example mission 5 represents a long duration transfer from an EM-L2 halo orbit to a region near the Sun-Earth (SE) L2 Lagrange point and then back to EM-L2. Figures XII and XIII show the round trip trajectory for this case in a SE Rotating Frame and a J2000-Earth Inertial Frame, respectively.

The Orion LEO to L2 mission phase to docking with the Core module at L2 is similar to that in mission 1, except that the return L2 halo target z-amplitude for this mission is 3,000 km. After a coast of about 2 days, the Core/Orion stack departs L2 to a 180-day roundtrip mission that takes the Core/Orion spacecraft beyond the SE-L2 point and back to EM-L2. The 237 m/s total ΔV cost for this sequence consists of 4 individual maneuvers: L2 halo departure, outbound mid-course correction, inbound mid-course correction, and L2 halo return arrival. After about 2 days in the L2 halo, the Orion separates from the Core module and performs a 944 m/s Earth return ΔV maneuver taking the crew to Earth entry interface in about 5.9 days. The Core module remains in the L2 halo. The ΔV and flight time summary for mission 5 is shown in Table VIII. The total mission duration is approximately 200 days (16 days in the Orion, and 184 days in the Core/Orion stack).

Maneuver	ΔV (m/s)
Earth Departure	3158
Lunar Flyby	228
L2 Halo Insertion	112
L2 to NR to L2	237
L2 Departure	944

Mission Phase	Flight Time (days)
LEO to L2 Halo	8.6
L2 Halo Rendezvous	1.5
Stay in L2 Halo	2.0
L2 to SE-L2 Region to L2	180.0
Stay in L2 Halo	2.0
L2 to Earth Return	5.9
Total	200.0

Table VIII: ΔV and Flight Time Summary for Mission Example 5.

IV. SUMMARY

The GER Asteroid Next missions to a near Earth asteroid represent tremendous challenges because of the nature of deep space travel. Longer missions, farther away from planet Earth, drive advancements in our current capabilities and knowledge. The missions described here represent potential options for conducting crewed missions of increasing complexity and duration during a risk reduction campaign prior to committing to a full-blown asteroid exploration mission. Utilizing missions of this nature would provide the opportunity for a step-wise approach for advancing the capabilities and knowledge ultimately required for long duration missions in deep space. These advancements ultimately are required for human missions to Mars, but could be initially be acquired relatively close to the safety of Earth.

V. Acknowledgments

The authors wish to acknowledge Dr. Juan Senent for his expertise and technical support on this task.

VI. References

[1] Global Exploration Roadmap, ISECG, September 2011, NP-2011-09-776-HQ.

[2] Farquhar, R. W., "The Utilization of Halo Orbits in Advanced Lunar Operations", NASA TN D-6365, July 1971.